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NASA TECHNICAL MEMORANDUM

NASA TM X-52622

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STALL AND DISTORTION INVESTIGATION OF A YTF30-P-1 TURBOFAN ENGINE

by John H. Povolny
Lewis Research Center
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TECHNICAL PAPER proposed for presentation at
Airframe-Propulsion Compatibility Symposium
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(NASA-TM-X-52622) STALL AND DISTORTION INVESTIGATION OF A YTF30-P-1 TURBOFAN ENGINE (NASA) 18 p

N74-77507

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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Abstract

Preliminary results on a YTF30-P-1 engine are presented from a program that was instituted at the NASA-Lewis Research Center to evaluate and understand the effect of both steady state and dynamic distortions on the performance and stall characteristics of advanced turbine engines. The high response compressor inter-stage instrumentation and specialized testing techniques that were required for the conduct of this program are described herein, and some of the data so obtained are presented. The preliminary results indicate that the instrumentation and specialized techniques do provide invaluable information that will help immeasurably in formulating engine dynamic simulations.

I. Introduction

Operational limits of propulsion systems may restrict the capabilities of subsonic and supersonic aircraft. One such limit of gas turbine propulsion systems is compressor stall or surge. The performance and operating limits of the engine, especially the compressor, are influenced by the flow conditions at the compressor inlet. This leads to the necessity of adequately matching the inlet and engine so that the capabilities of the aircraft are not compromised by the flow condition supplied by the inlet and/or the susceptibility of the engine compressor to the inlet flow condition.

A program to evaluate and understand the influence of both steady state and dynamic inlet disturbances on the operating limits of advanced turbine engines has been initiated at the NASA-Lewis Research Center. The objectives of this program are presented in the first figure. The first engine to be investigated in this program was a YTF-30 P-1 afterburning turbofan engine which was selected as being typical of those used or proposed for supersonic military aircraft. To accomplish the desired evaluation and understanding it was necessary to develop a high response measurement capability for pressures within as well as at the inlet and exit of the compressors. Such capability is required in order to determine the location of the occurrence of stall and its progression through the compressor. It was also necessary to develop and evaluate some specialized testing techniques that would either provide the desired environment or would initiate the compressor stall in a desired location. In the interest of brevity, this paper will present a synopsis of this work to date, while hitting some of the high spots. An NASA report (ref. 1) has been issued on the first phase of this program; it covers a compressor stall evaluation of the YTF30-P-1 engine with no inlet distortions. A second report covering the effects of steady state distortions on the performance of this engine is currently underway, and the data for a report on the effects of dynamic distortions is currently being analyzed. Some typical data from each phase of the program will be presented herein.

II. Engine Instrumentation

High response compressor pressure instrumentation was installed at the fan inlet (sta. 2.0), first stage stator (sta. 2.1) fan tip outlet (sta. 2.3f), fan hub outlet or low compressor inlet (sta. 2.3), sixth stage stator (sta. 2.6) low compressor outlet (ninth stage) or high compressor inlet (sta. 3.0), twelfth stage stator (sta. 3.12), and high compressor outlet (sixteenth stage and sta. 4.0). There were also high response pressure measurements in the combustor, fan duct and afterburner, as well as conventional steady state instrumentation throughout the engine. Capability was provided in the fan inlet rakes for a total of 40 high response total pressure measurements but this total was utilized only where required because of data acquisition limitations. All of the other compressor high response instrumentation consisted of two pairs of total and static pressures located approximately 180^{o} apart installed at each of the stations indicated above. The response of these probes was linear within $\pm 5\%$ out to about 500 cps. A sketch of a typical interstage probe is shown in figure 2. Miniature transducers, 1/4 inch in diameter, were used and were located as close to the probe inlet as possible ($<$ 2 inches). Inasmuch as the transducers were limited to an operating temperature of 170^{o}F (the gas temperature was higher than this at many of the stations) and their calibrations were sensitive to temperature, it was decided to water cool all of them so that they would be at a low uniform temperature. The transducers were thus mounted in a water cooled jacket, the individual assemblies of which could be removed for repair without having to remove the rake or probe from the engine. In spite of the aforementioned precautions it was determined from preliminary calibration of these transducers that zero and sensitivit shifts occurred as a function of time. Therefore a system was devised to calibrate these transducers in their operating environment inside the engine just prior to recording data. Details of this calibration system and the instrumentation are presented in the reference 1. The high response data were recorded simultaneously on a high speed digitizer recorder and on magnetic tape. Inasmuch as the digitizer had a sampling rate of only 60 samples per second, the digitized information was used just prior to stall; the actual stall point and sequence of events during stall were obtained from the magnetic tape analog traces.

III. Engine Installation

The engine installation in the altitude chamber was a conventional direct connect type. The altitude chamber included a forward bulkhead which separated the inlet plenum from the test chamber. Conditioned air was supplied to the plenum at the desired pressure and temperature. The chamber aft of the bulkhead was evacuated to the desired altitude pressure. The conditioned air flowed from the plenum through a bellmouth and inlet duct to the engine. A high response bypass valve located in the bulkhead allowed some air to bypass the engine and was automatically controlled to maintain a constant inlet pressure and ram pressure ratio across the engine during both steady state and transient engine operation. The exhaust from the engine was captured by a collector extending through a rear bulkhead and discharged into the facility exhaust system wherein the exhaust pressure was maintained constant by automatically controlled valves.

IV. General Test Procedure

The investigation was conducted at a Reynolds number index of 0.5 with an inlet temperature of 60°F (16°C). This resulted in an inlet total pressure of 7.5 psia (5.2 N/cm^2). Ram pressure ratio across the engine was held constant at 3:1 to assume a choked exhaust nozzle for all operating conditions. These test conditions were selected for several reasons. First, this Reynolds number index corresponds to a range of flight conditions representative of the flight regime of a supersonic aircraft, and second, the loss in performance due to Reynolds number effect was not expected to be significant at this level. Also, the pressure loadings on the instrumented compressor casing would be lower than at sea level pressure especially during stall and compressor surge.

In the initial stages of the program when operating the engine without inlet distortions two techniques were employed to induce stall in the three compressor units (fan, low compressor, and high compressor). The low pressure compressor operating line was raised toward the stall limit by increasing the exhaust nozzle area above rated; the engine was then very slowly accelerated (10 to 20 rpm per second) until the engine either stalled or reached one of its operating limits. In a similar manner the exhaust nozzle was closed to raise the operating lines of the high compressor and fan unit, and again a slow acceleration was made until stall occurred in one of these units or an engine operational limit was reached. The slow acceleration was used to keep the engine close to the steady state operating line for the particular exhaust nozzle area being employed. Data from the transient instrumentation were recorded during these slow accelerations until sometime after the engine had stalled. The second method of inducing stall (in the high compressor) was to inject an increment of fuel into the engine fuel system, which increment would cause a step increase in pressure at the high compressor exit. The size of the fuel step was then successively increased until stall was encountered in the high compressor. Stall data were obtained in the aforementioned manners over a range of rotor corrected speeds from idle to maximum. In addition to the foregoing there were some high response stall detection, overspeed and overtemperature systems that were employed to protect the engine during stall; these systems are also described in reference 1. There were also some newly developed techniques employed for stalling the engine and creating distortions which will be described in a subsequent section.

V. Clean Inlet Stall Data

Analysis of the transient data to obtain stall pressure ratios and rotor corrected speeds for each stage group and for each compressor component (fan, low, and high) was done in two steps: (1) identification of the stage group and component that stalled first; and (2) determination of the pressure ratio and corrected speed for that group and component immediately prior to stall. Identification of the unit stalling first was made from the analog total pressure time histories. A drop in the discharge and a corresponding rise in the inlet pressure of the group of stages or component identified it as stalling first. Pressures falling downstream and rising upstream of the stage group also aided in the identification. The pressure ratio and corrected speed of the stalling unit were taken from the computed output of the high speed digital system when stalls were obtained with slow accelerations. For the fuel step transient data, the rate of change of pressure was too great for the digital system and the engineering data were computed from the analog traces.

In some cases a method of analysis was used (ref. 2) for determining the stall pressure ratio of stages ahead of the stage that stalls initially. This method is based on the fact that as the first stage stalls its flow is reduced so rapidly that the pressure at the exit of the preceding stage will rise before the inlet pressure of that stage is affected. Thus, this stage in turn stalls and the process is repeated until the stall has progressed completely through the compressor to the inlet. The fuel step technique usually initiates stalls in the downstream stages of the high compressor, and thus with this technique the stall pressure ratio of all stage groups can be determined from a single transient.

Typical time histories of pressures in the compressors for three types of observed stalls are presented in figures 3, 4, and 5. Figure 3 shows a slow acceleration to a low compressor stall with a large exhaust nozzle area (186% of rated); figure 4 shows a slow acceleration to a high compressor stall with a small exhaust nozzle area (78% of rated); and figure 5 shows a typical fan rotating stall pattern which often followed a hard stall in either of the other compressors. The vertical lines on each curve are for time correlation and each interval represents 0.01 sec. Figure 3 (low compressor stall) shows a rise in $P_{T,2.6}$ (sixth stage) and a drop off in P_3 (ninth stage) occurring shortly before the abrupt changes in the other pressures indicating the initiation of stall in the group consisting of stages 6 through 9. The progression of stall forward from station 2.6 (sixth stage) to 2.3 (third stage), 2.1 (first stage), and finally 2.0 (inlet face of engine) is also shown. Figure 4 (high compressor stall) shows a simultaneous rise in $P_{T,3.12}$ (twelfth stage) and a drop in $P_{T,4}$ (sixteenth stage or compressor exit) indicating that the stall originated between those locations. The progression of the stall forward through the compressors can be seen in this case, also. Figure 5 (fan rotating stall) indicates that a single zone stall pattern rotating at 43 percent of low rotor (fan) speed was occurring. The 180° phase shift indicated by the two $P_{T,2.3}$ traces which were approximately 180° apart, confirm the one zone pattern. The pressure rise at station 2.1 (stage one) and the simultaneous drop off at station 2.3 (third stage) indicate that the stall was originating somewhere between the first and third stages. The drop off in pressure in all of the succeeding stages indicates that its effect was felt all the way through the low and high compressors.

The effect of the rotating stall on engine operation is presented in figure 6 which shows that the engine could not be accelerated above a high rotor speed of about 11,800 rpm while in rotating stall. This was due to a shift in the operating line of the engine which intersected the fuel control acceleration schedule at about 10,800 rpm thus preventing any further acceleration. In the interest of brevity none of the compressor and fan maps (with stall lines) that were obtained in this program will be presented (they are available in ref. 1). Suffice it to say, however, that in this program, as in other similar programs, there were determined to be discrepancies between compressor rig data and compressor data obtained on the engine. It appears that at least a part of this discrepancy is a result of the different environment that the compressor is exposed to on the engine as compared to the rig and this difference may be classified as a distortion effect. For example, the low compressor discharge velocity, temperature, and pressure profiles which are fed into the high compressor may be completely different than those in the rig test and could have a very significant effect on its performance.

VI. Special Techniques and Results

Four special testing techniques that have been or are about to be employed in this program are indicated in figure 7. The pressure jet distortion system has been employed on the TF-30 engine to produce both steady state distortions and fluctuating pressures in the inlet duct ahead of the engine. The engine effects obtained when producing steady state distortions with the jets were practically identical to those obtained when using screens but were much easier to run off. Some of the preliminary results obtained when using the jets to produce fluctuating pressures will be discussed in succeeding paragraphs. The temperature distortion device has not been used to date but it has been built and is about to be checked out. It consists of a large (5½ ft. diameter) hydrogen burner mounted in front of the engine inlet bellmouth and has the capability of producing steady state circumferential and radial distortions, or providing temperature ramps as high as 5000°F per second. The choked inlet duct also has not been run on the TF-30 engine but will be shortly. It is similar to the device used by Kimzey (ref. 3) and has the possibility of simulating the environment provided by an aircraft inlet. This possibility is one of the items that will be checked out when the choked inlet is employed. The fourth technique involves reversing the flow in the compressor discharge ports and hence the name "compressor discharge in-flow bleed." It, of course, requires an external source of high pressure air but has the advantage of being able to "sneak up" on the stall limit of the high compressor at high corrected speeds without over temperaturing the turbine.

A photograph showing the arrangement of the pressure jet system jet nozzles in the engine inlet duct is presented in figure 8, and a photo showing an external view of the engine installation with the pressure jet system installed is presented in figure 9. In this system secondary jets of air are injected counter to the primary air flow in the engine inlet duct forward of the compressor face. Through control of the secondary-air distribution and flow rate, variable amplitude steady state or dynamic pressure distortions or uniform dynamic pressure oscillations can be produced. As can be seen in figure 8, the secondary jet nozzles are uniformly distributed circumferentially and radially in a pattern that repeats every 60 degrees of circumferential extent. Six high response servo-operated valves, designed especially for this application, are employed to control secondary air flow to each 60-degree segment. Separate flow lines are provided from each control valve to each jet nozzle in the segment with all lines having equal cross-sectional area and length. Momentum interchange between the secondary and primary streams occurring upstream of the jet nozzle array is primarily responsible for the total pressure loss that is generated. The amount of total pressure loss incurred is controlled by varying the secondary air flow. If it is varied in a pulsating manner then a pulsating pressure is generated at the engine face.

The performance of the pressure jet distortion system when operated in the pulsating mode (all jets operating) is summarized in figure 10 wherein the ratio of engine inlet total pressure amplitude to engine inlet steady state (average) pressure is plotted against the frequency of the pressure oscillation. It can be seen that the amplitude is relatively constant at a value slightly greater than 0.3 out to a frequency of about 20 HZ at which point it rapidly falls off to a value of slightly less than 0.10. It would be desirable to maintain the amplitude at the higher value out to as high a frequency as possible and some work is currently being performed toward that end. Preliminary results indicate that it should be possible to achieve an amplitude of about 0.20 at a frequency of 100 HZ. A plot of some of the individual pressure measurements at the engine inlet face as a function of time is presented in figure 11 for a pulsation frequency of 10 HZ. It can be seen that all of the individual pressure measurements are very well in phase and close together in amplitude and that a reasonable approximation to a sine wave was obtained.

Some of the preliminary engine performance data obtained when pulsating the inlet at a frequency of 10 HZ are presented in figures 12 and 13. Figure 12 presents the variation of a number of compressor interstage pressures as a function of time for an average steady state inlet total pressure of 7.4 psia and 87 percent corrected speed. The significant items to note on this curve are that the pressures measured at the high compressor discharge (sta. 4) and fan tip discharge (sta. 2.3F) appear to be more attenuated and seem to have more of a phase shift with respect to the preceding stage measurement than any of the other measurements. It is interesting to note that in each of these two cases there is a relatively large volume behind the station in question (combustor volume behind sta. 4 and fan duct volume behind sta. 2.3F). If the pressure ratios across the various stage groups are computed from the data in figure 12 and plotted in a similar manner against time the resulting curves presented in figure 13 are obtained. Here again the significant pressure ratio curves are for the stage groups immediately adjacent to a large volume (stage group 2.0 to 2.3F and stage group 3.12 to 4.0). It is to be noted that not only do these two stage groups go through a much greater pressure ratio variation but their swing above their steady state value of pressure (indicated by the short horizontal line on the right and measured just prior to the pulsation) is significantly greater than below it. The data presented in figures 12 and 13 were obtained at an amplitude just slightly less than that required to stall the compressor and is considered to be typical of conditions just prior to stall. Examination of the analog traces at the stall point indicated that the stall originated in the back end (stages 12-16) of the high compressor even though the pressure ratio excursion across the fan tip (stages 1-3F) was greater. Apparently the fan has more stall margin built into it than the high compressor.

If the ratio of peak pressure ratio to steady state pressure ratio for the various stage groups is plotted against frequency for some of the data obtained to date the resulting curves presented in figure 14 are obtained. Here again, the interesting curves are for the 2-2.3F and 3.12-4.0 stage groups. The data represented by the solid lines were obtained for an inlet amplitude of about 0.28

and are representative of performance just prior to stall. The data represented by the dashed curve are extrapolations of trends obtained at a lower inlet amplitude and are probably representative of the performance that might be obtained if the engine did not stall at the higher amplitude.(The higher amplitudes weren't obtainable at the higher frequencies.) In addition to analyses, similar to those described, that are currently under way the results of this investigation (in the form of generalized stage group performance) are also being utilized in formulating an engine dynamic simulation.

VII. Concluding Remarks

Preliminary results of a program to investigate the stall and distortion characteristics of a turbofan engine indicate that:

(1) Data from detailed high response compressor interstage instrumentation can provide an insight into the stall mechanism, its origin and progression through the compressor system, and the compressor interaction with other engine components.

(2) It is possible to evaluate in an engine the static and dynamic performance of each compressor component, stage group or even individual stages by the employment of specialized techniques, described herein.

(3) Results obtained on a component in an engine are not necessarily the same as those obtained in a component rig because of differences in inlet environment and component interactions.

(4) Information obtained from the type of program described herein will be invaluable in formulating and verifying the dynamic simulation of a turbofan engine both with and without inlet air distortions.

References

1. Braithwaite, Willis M.; and Vollmar, William R.: Performance and Stall Limits of a YTF30-P-1 Turbofan Engine with Uniform Inlet Flow. NASA TM X-1803, 1969.

2. Lubick, Robert J.; and Wallner, Lewis E.: Stall Prediction in Gas-Turbine Engines. J. Basic Eng., vol. 81, no. 3, Sept. 1959, pp. 401-408.

3. Kimzey, W. F.: An Investigation of the Effects of Shock-Induced Turbulent Inflow on a YJ93-GE-3 Turbojet Engine. ARO, Inc. (AEDC-TR-66-198, DDC No. AD-377312L), Nov. 1966.

CURRENT FULL SCALE ENGINE PROGRAM OBJECTIVES

1. EVALUATE STALL AND DISTORTION CAPABILITIES OF A HIGH MACH NO. AFTERBURNING TURBOFAN OVER THE RANGE OF FLIGHT MACH NO.
2. PINPOINT THE ORIGIN OF THE STALLS AND DEFINE THEIR PROGRESSION THROUGH THE MACHINE
3. DEFINE THE DYNAMIC INTERACTIONS OF THE VARIOUS COMPONENTS SUCH AS THE COMPRESSORS, AFTERBURNER, COMBUSTOR, AND INLET
4. FORMULATE AN ENGINE SIMULATION USING THE INFORMATION FROM ITEMS 1, 2, AND 3 THAT WILL ACCURATELY PREDICT ENGINE PERFORMANCE FOR ALL FLIGHT ENVIRONMENTAL CONDITIONS THAT COULD BE ENCOUNTERED
5. WORK OUT SOLUTIONS TO BASIC ENGINE PROBLEMS THROUGH UNDERSTANDING RATHER THAN TRIAL AND ERROR

Figure 1.

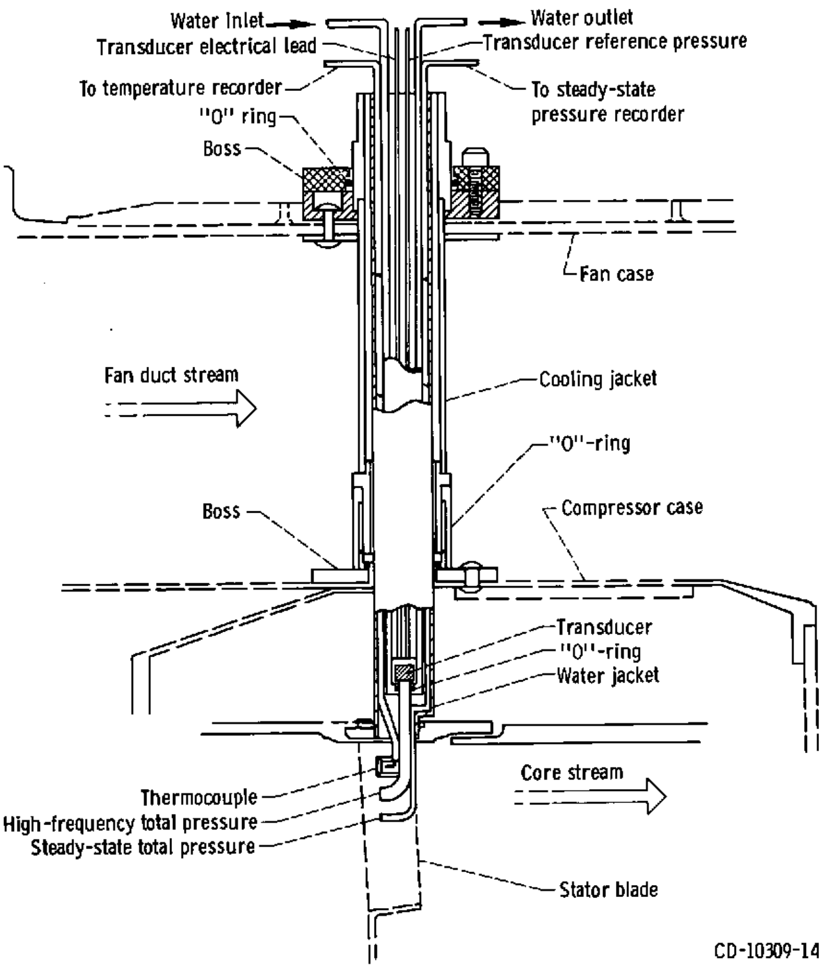
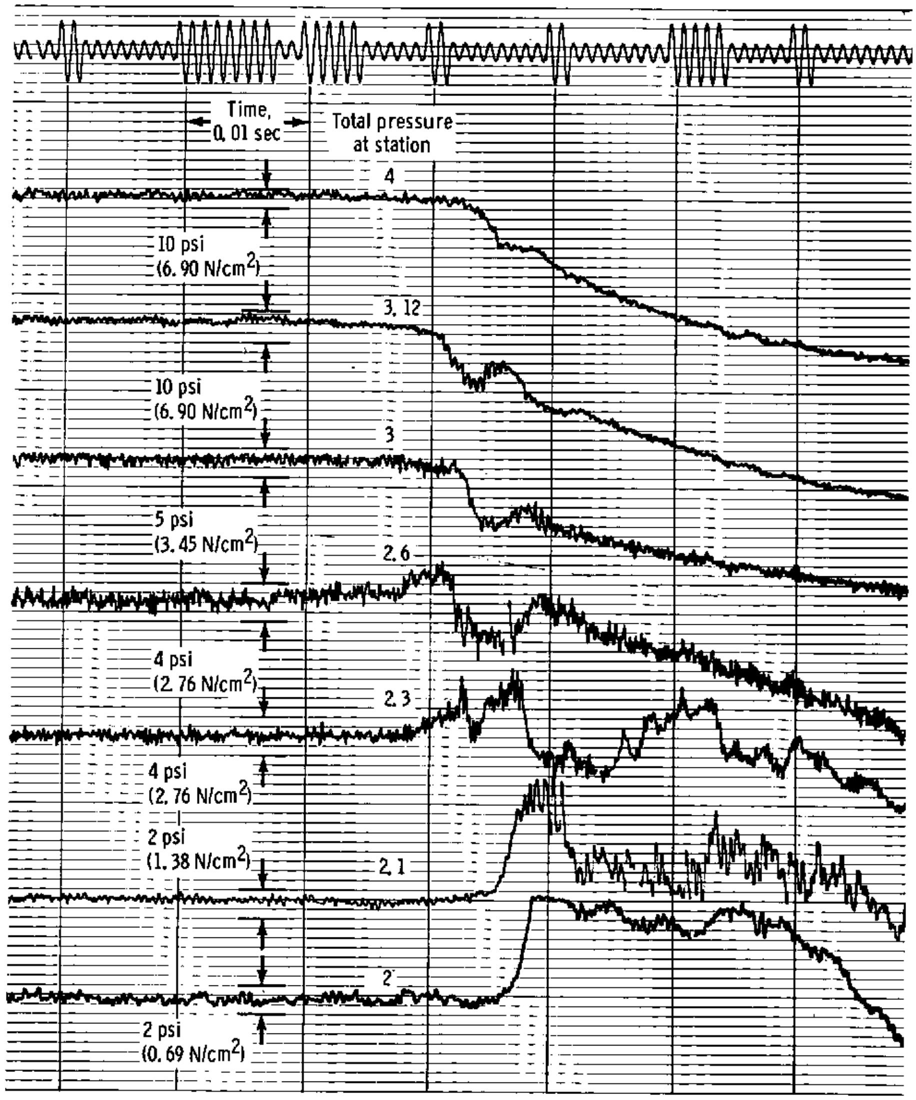
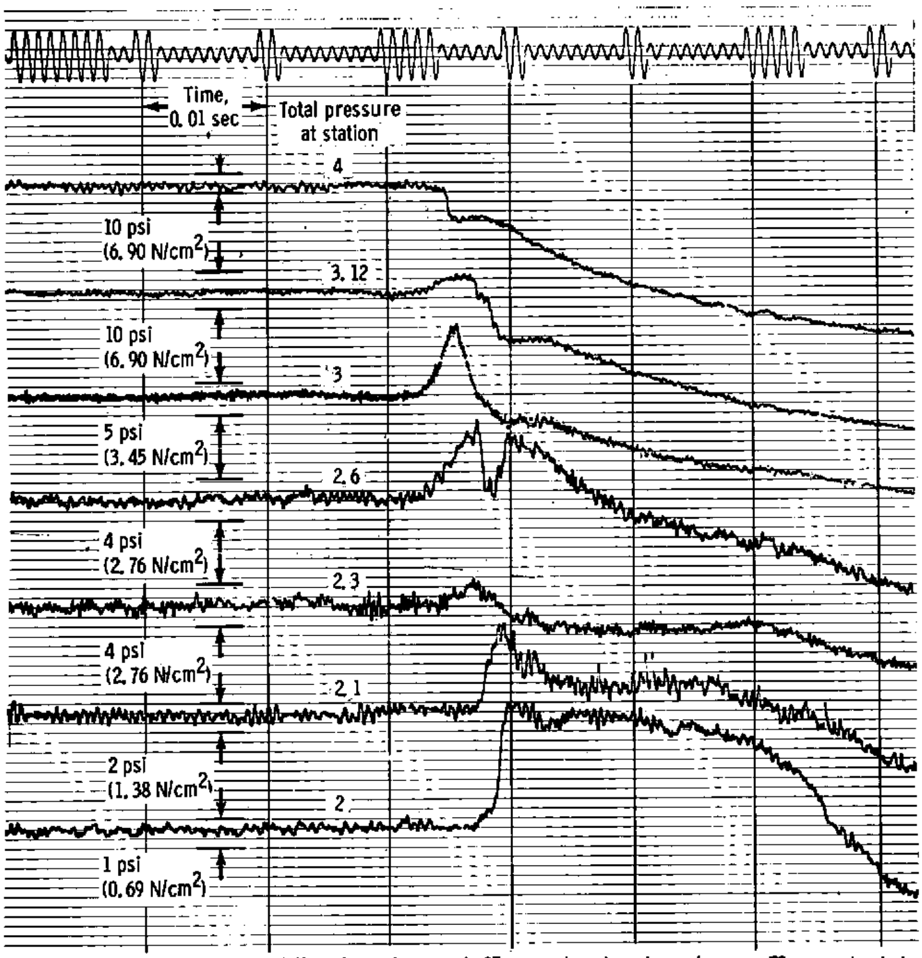


Figure 2. - Typical probe installation.



Low-pressure compressor stall. Low rotor speed, 73 percent; exhaust nozzle area, 186 percent rated.

Figure 3. - Typical time histories during stall transient of YTF30-P-1 turbofan engine at Reynolds number index of 0.5.



High-pressure compressor stall. High rotor speed, 97 percent; exhaust nozzle area, 78 percent rated.

Figure 4. - Typical time histories during stall transient of YTF30-P-1 turbofan engine at Reynolds number index of 0.5.

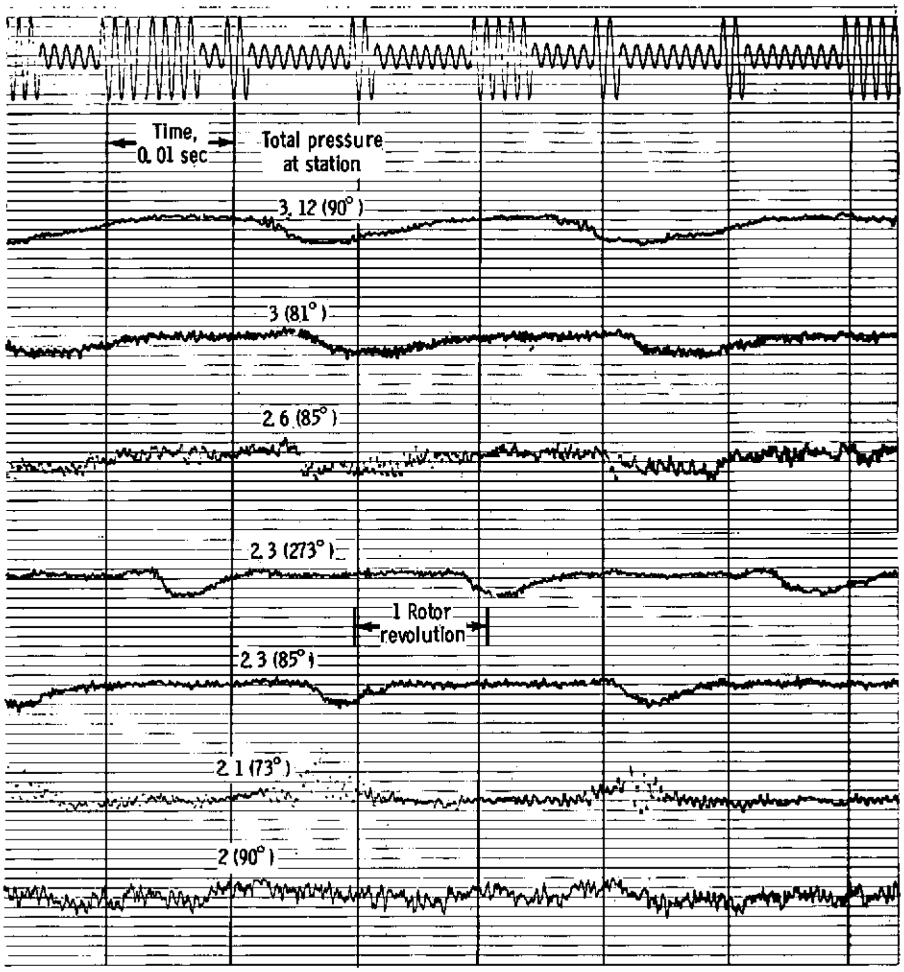


Figure 5. - Typical rotating stall pattern observed in YTF30-P-1 turbofan engine following compressor stall. Low rotor speed, 56 percent, high rotor speed, 77 percent; exhaust nozzle area, 186 percent rated. Reynolds number index, 0.5; stall frequency, 0.43 low rotor frequency.

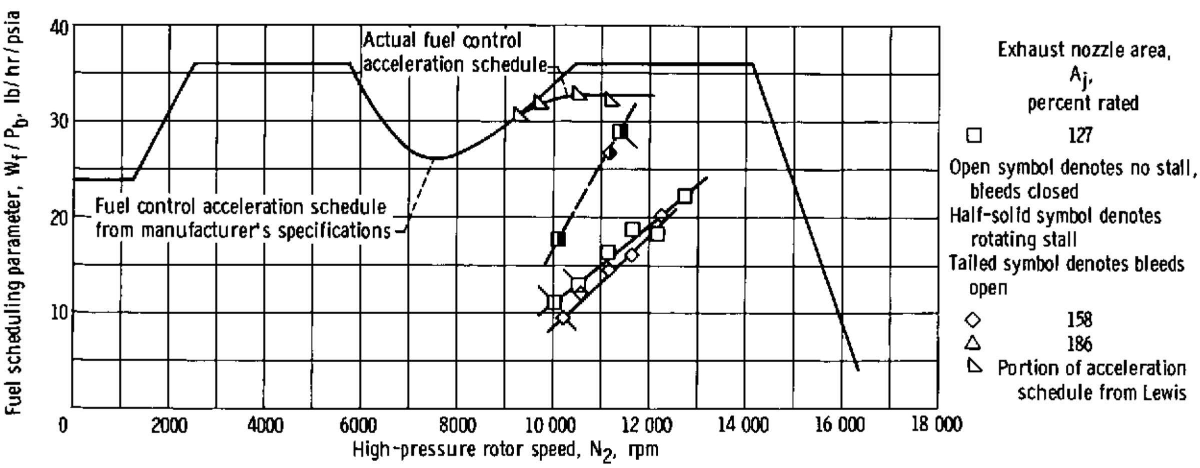


Figure 6. - Fuel schedule for YTF30-P-1 turbofan engine. Total temperature at station 2, 59° F (15° C).

1. PRESSURE DISTORTION JETS
2. TEMPERATURE DISTORTION DEVICE
3. CHOKED INLET DUCT
4. COMPRESSOR DISCHARGE IN-FLOW BLEED

Figure 7. - Four special testing techniques used to determine engine transient and stall characteristics.

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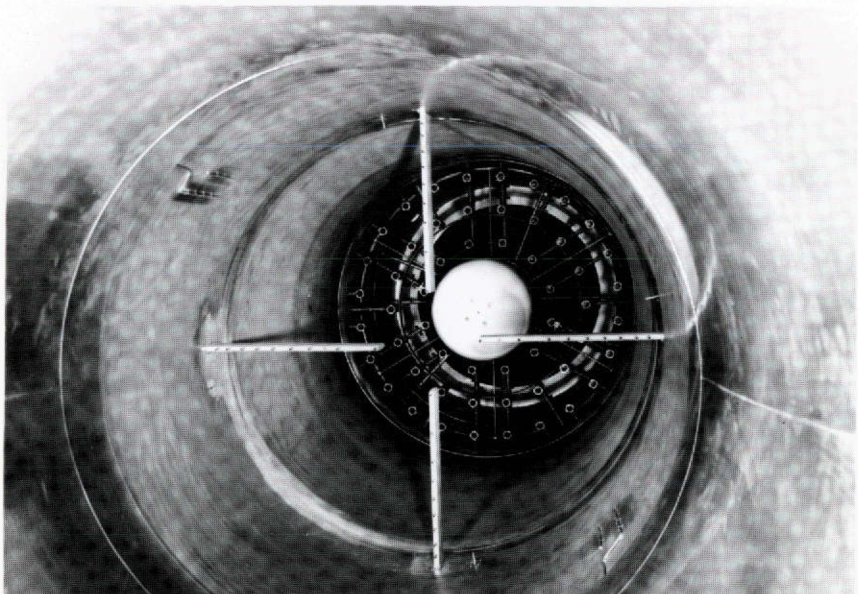

Figure 8. - Pressure jet stream installed in engine inlet duct (view looking toward engine inlet).

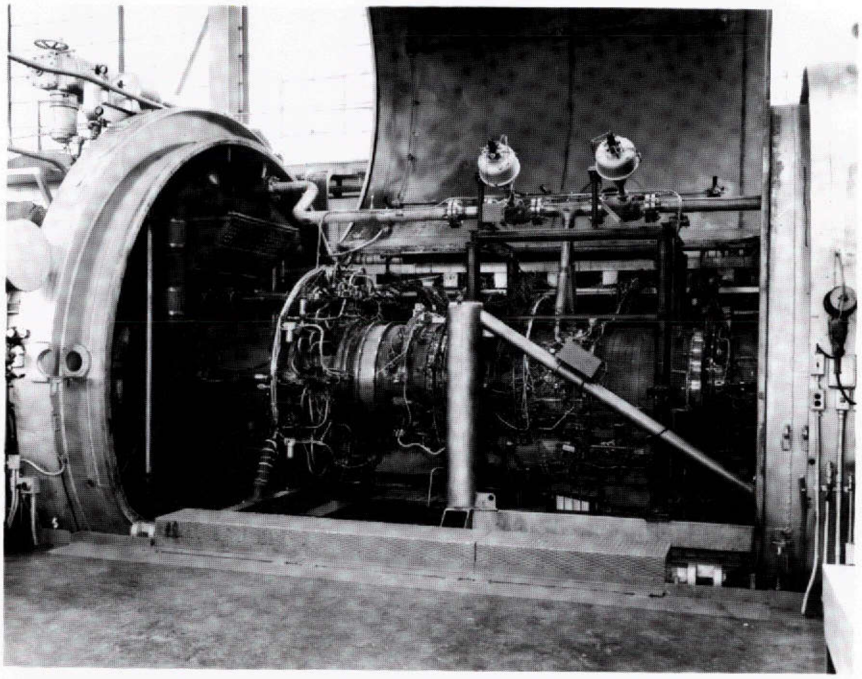

Figure 9. - Pressure jet system installed (external view of engine inlet duct).

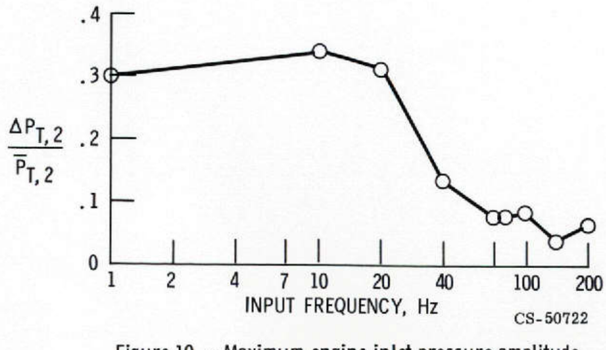

Figure 10. - Maximum engine inlet pressure amplitude over a range of input frequencies.

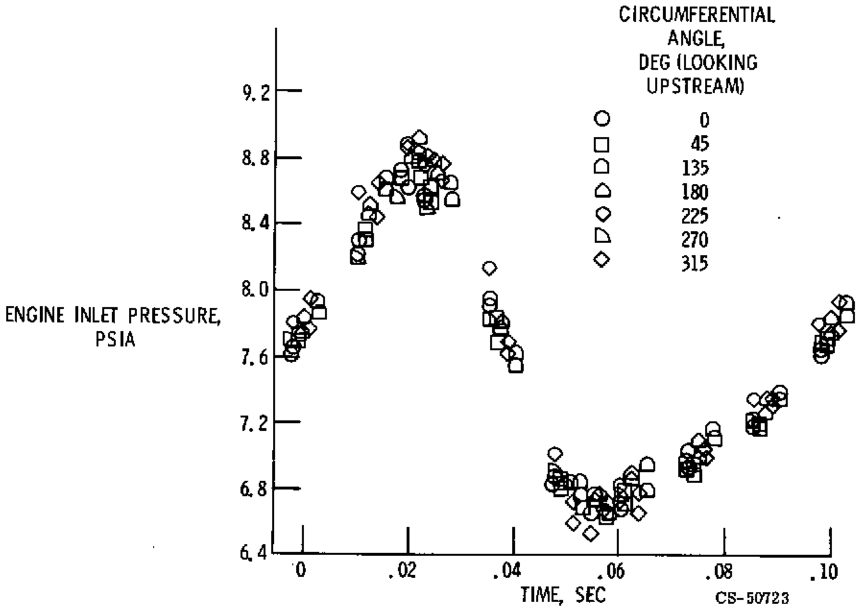

Figure 11. - Variation of engine inlet pressure with time. Input frequency, 10 Hz.

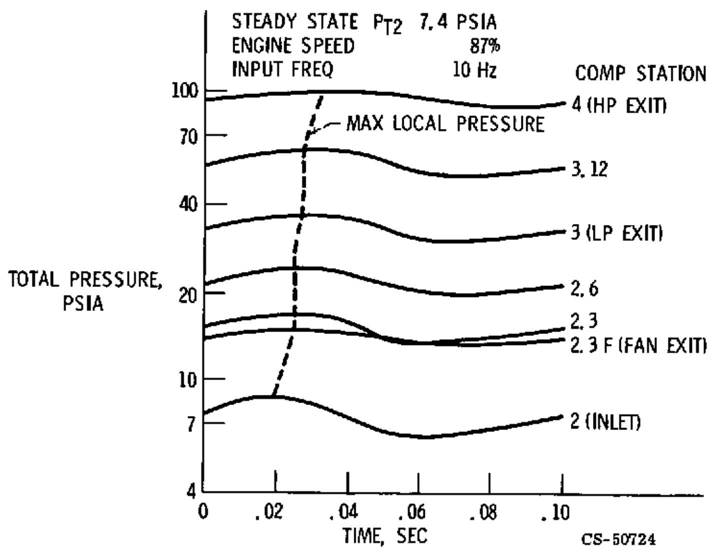

Figure 12. - Variation of several compressor interstage pressures during uniform engine inlet pressure variations.

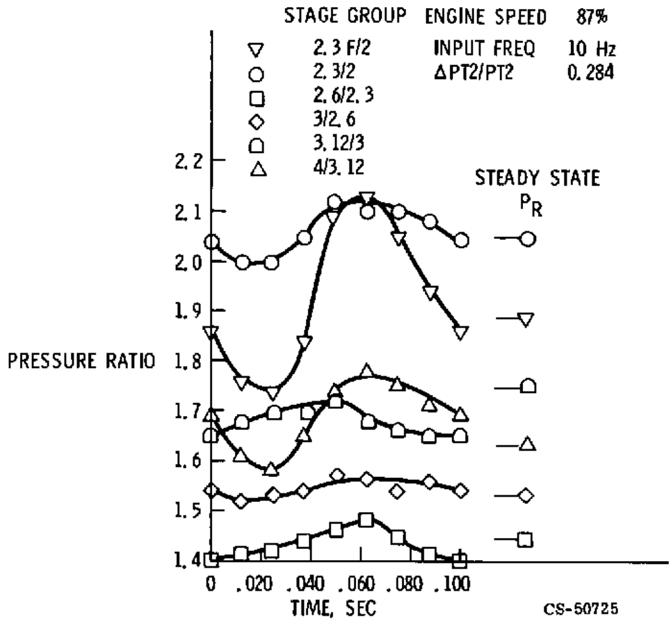

Figure 13. - Variation of stage group pressure ratios during uniform inlet pressure variations.

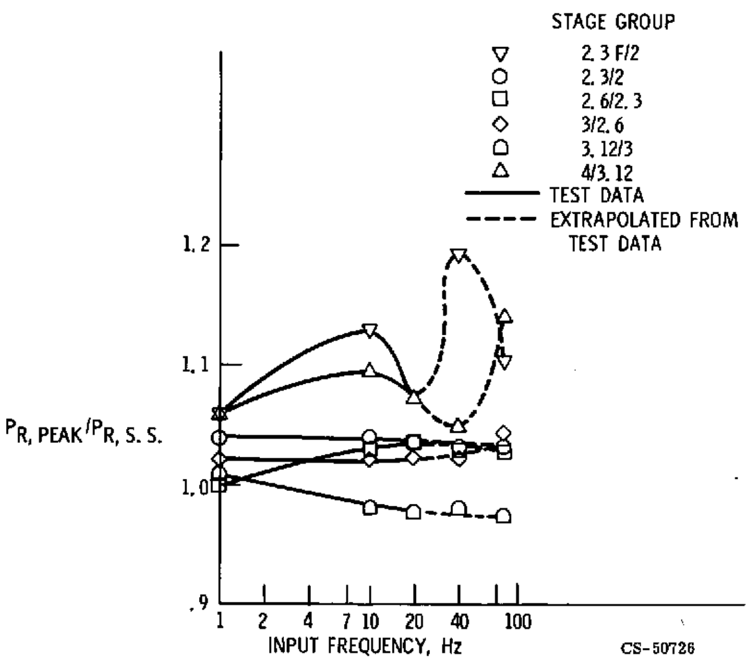

Figure 14. - Variation of peak pressure ratio with frequency, $\Delta P_{T2}/P_{T2}$, 0.28.